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ANALYSIS OF A CRYOGENIC SERVICE MODULE FOR THE APOLLO MISSION

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ANALYSIS OF A CRYOGENIC SERVICE MODULE FOR THE APOLLO MISSION

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SUMMARY

A preliminary analysis of a cryogenic service module for the Apollo lunar landing mission has been carried out. The chosen design is based on a hydrogen-oxygen pump-fed system utilizing a single RL-10 engine. A cold helium pressurization system is assumed with sufficient capacity to provide multiple restart capability. At the present time a pressurized service module is being developed with a propellant capacity of 45,000 pounds which could utilize a lunar transfer weight up to 105,000 pounds. It is estimated herein that the present service module could provide a LEM gross weight of 30,080 pounds for a lunar transfer weight of 90,000 pounds.

Development of a cryogenic service module with a propellant capacity of 33,000 pounds could also utilize a lunar transfer weight up to 105,000 pounds. Such a stage when off-loaded could provide a LEM gross weight of 40,000 pounds for a lunar transfer weight of 90,000 pounds. This represents a 33 percent increase in LEM gross weight over that for the present service module. The increased capability of the cryogenic service module can also be exploited to reduce the required lunar transfer weight. For example, a LEM gross weight of 30,080 pounds will require a lunar transfer weight of less than 80,000 pounds when the cryogenic service module is used in the mission. Similarly, combinations of improvements in both LEM weight and lunar transfer weight requirements could be obtained.

A smaller cryogenic service module more nearly optimized for the present LOR mission would have a propellant capacity of approximately 30,000 pounds. Such a design exhibits only small performance improvements compared with the larger cryogenic service module, but it might achieve inherent advantages in the suppression of propellant slosh during boost and in increased volumes available for necessary auxiliary equipment.

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INTRODUCTION

Recent interest in performance improvements for the Apollo lunar landing mission has focused attention on the concept of a cryogenic service module. Accordingly, the Lewis Research Center has directed a small analytical effort toward a study of a cryogenic service module in an effort to arrive at an estimate of the possible benefits which might be realized from such a development. Previous Lewis studies have been directed toward lunar landing modules for various Apollo mission concepts and toward a study of a lunar logistic spacecraft (References 1-3). Reference to the previous studies, together with preliminary analysis in the present study established to greater capability of a pump-fed hydrogen-oxygen service module compared with a pressure-fed hydrogen-oxygen design. This, together with the satisfactory developmental status of the RL-10 engine has led to the adoption of a pump-fed design for this study. To obtain maximum performance improvements the dimensions of the cryogenic service module were not constrained in the envelope of the present service module. An entirely new service module design was assumed, with overall dimensions somewhat larger than the present design. Attention has been focused on a design utilizing a cold helium pressurization system and a single RL-10 engine. The engine as presently constituted would require additional development for the cryogenic service module application. Reliable multiple restart capability under vacuum conditions would have to be demonstrated with emphasis on engine start and operation at low NPSH. In addition, developmental effort in the areas of valve leakage rates, chill-down requirements and repeatability of start and stop transients would be required.

The design concepts adopted in this study are intended to be consistent with the realization of a cryogenic service module of high reliability. However, more design effort will be necessary to establish the details of the final configuration and ultimately the final performance and reliability of the cryogenic service module design.

The study has been directed toward an analysis of the lunar orbit rendezvous mode as it would be accomplished with a single Saturn V launch. Some details of the assumed mission profile together with a description of the configuration adopted in the analysis are summarized in this report.

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MISSION PROFILE

The planned mission profile followed herein utilizes the Saturn V boost vehicle with an Earth-orbit parking mode and a lunar-orbit rendezvous (LOR) mode. A synchronous descent maneuver from the service module orbit altitude for the Lunar Excursion Module (LEM) is assumed with rescue and retrieval capability of the LEM by the service module provided. A total of eight midcourse corrections (four outbound and four inbound) are made using either the service module main propulsion system or the service module attitude control engines. The main propulsion system is used for three midcourse corrections, retro into lunar orbit, LEM rescue and retrieval and departure from lunar orbit.

In the example mission examined in this study, an Earth orbit stay time of 4 1/2 hours is assumed followed by injection to lunar transfer using the SIVB stage. The first two midcourse corrections are made at 9 and 15 hours after injection using the service module main propulsion system. The next two corrections are made at 44 and 69 hours after injection using the service module attitude control engines. Retro into an 80 nautical mile lunar orbit occurs 72 hours after injection. Two orbits of the spacecraft are completed prior to LEM descent. Possible rescue would be accomplished by placing the main spacecraft on the same orbit as the LEM and would occur on the first opportunity after initiation of the LEM synchronous descent. Use of the main propulsion system is planned for the rescue maneuver. A total stay time in lunar orbit of 48 hours is allowed. Insertion on to the moon-earth transfer trajectory occurs five days after departing from Earth and again uses the RL-10 engine system. Four midcourse corrections are used on the return trip and occur at 9, 15, 44 and 69 hours respectively following insertion. The midcourse 9 hours after departure from lunar orbit uses the main propulsion system; the following three corrections use the service module attitude control engines. Earth re-entry utilizes atmospheric braking and is accomplished on the eighth day after launch. The velocity increments assigned to each propulsion phase are tabulated in Table I. Total velocity requirements are consistent with propellant requirements listed in Reference 4, and are intended to include a ten percent ΔV reserve.

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PRELIMINARY DESIGN

In arriving at preliminary design concepts and vehicle weight estimates, conservative assumptions were adhered to. Although no reliability analyses are available at the present time, it is apparent that the design should strive for simplicity and a minimum number of components. Consequently, simple direct approaches to the various design requirements were imposed; where this resulted in weight penalties and consequent performance deterioration, these were accepted without attempting to regain performance by compromising the basic design philosophy. In areas where little information is available at the present time, the assumptions may be relaxed when more data is acquired. Major design assumptions are summarized in Table II.

Materials and Design Criteria - Structure was designed for flight loads obtained from the Saturn V Launch Vehicle Design Data Book. Limit load factors of 1.1 on yield strength or 1.4 on ultimate strength at appropriate material temperatures were adhered to. All buckling loads were assumed to be ultimate. All structures and tanks were designed so that no yielding occurs at limit loads. Materials were selected on the basis of compatibility, strength, notch sensitivity, fracture toughness, assumed resistance to meteoroid impact, and forming and welding properties.

Titanium alloy (5Al, 2.5 Sn) was selected as helium tank material. With hydrogen-oxygen propellants, this vessel will be immersed in the liquid hydrogen tank. Based on fracture analysis, a stress of 120,000 psi was allowed at a temperature of 38°R. However, data on the plane strain fracture toughness of this material in thick sections is needed to demonstrate the validity of this working stress. The relatively small size of the helium tank will simplify quality control and inspection techniques. Furthermore, meteorite impingement will not occur unless a catastrophic penetration of the liquid hydrogen tank has already occurred.

A ductile aluminum alloy, 2219 — T87 was selected for the fuel and oxidant tanks. This material is very tough at cryogenic temperatures and should therefore offer high resistance in the presence of cracks or meteoroid damage. It is compatible with the oxidants and fuels considered. Minimum aluminum skin gages of 0.015 inch for the 68 inch diameter LOX tanks and 0.020 inch for the 167 inch diameter hydrogen tank were set as manufacturing and handling limitations. Pressure requirements alone lead to tank thicknesses less than these minimums. Actual propellant tank weight is not a large portion

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of total hardware weight for the pumped system.

Airframe - The outer shell of the airframe is made of aluminum honeycomb sandwich panels reinforced by eight longitudinal stiffeners and five circumferential rings. The sandwich panels are built of .016 inch thick, 7075 alloy face sheets bonded to 0.75 inch thick, 5052 alloy core. This core consists of foil sheets, .001 inch thick, expanded into honeycomb cells having a width of 3/16 inch across the flats. Longitudinal stiffeners made of 7075-T6 aluminum serve to distribute concentrated axial inertial loads into the sandwich panels. These loads are introduced by the spider beam and the LOX tank rings in the aft section of the vehicle and by eight command module support pads and fuel cell components in the forward section. The longitudinal members vary in size along the length of the vehicle according to the magnitude of the applied loads. Rings made of 7075-T6 aluminum resist radial inertial loads imposed upon the outer shell and are sized accordingly. Details of the outer shell structure are shown in Figure 2 b&c.

All of the main propellant tanks are supported by thin Titanium 5 Al - 2.5 Sn shells of revolution in order to avoid imposing concentrated loads or bending moments into the thin walled tanks and to prevent excessive heat conduction. The aft end of each support shell is mechanically fastened to the outer leg of a 2219-T87 "Y" ring which has been welded to the spherical tank bulkheads, figure 2b. The forward end of the hydrogen tank support cone is mechanically fastened to a 7075-T6 aluminum ring attached to the outer shell. The forward end of each oxygen tank support cylinder attaches to a 7075-T6 aluminum box ring which, in turn, is supported at three points, one point at the tangency of each adjacent spider beam and one point at the outer shell ring. The cruciform spider beam built of 7075-T6 aluminum also transmits engine thrust and inertia loads into the outer shell. The helium tank is supported by a titanium 5 Al - 2.5 Sn conical shell mechanically fastened at the aft end to an aluminum ring attached to the tank bulkhead. Circular cutouts in this shell are provided for propellant flow. Doublers in the aft bulkhead are used to reinforce the bulkhead and distribute helium tank inertial loads.

The aft bulkheads of the LOX tanks are shielded from meteoroids and engine heat with 0.02 inch thick laminated Fiberglas bonded to one inch of polyurethane foam. This insulation is in addition to the foil insulation provided for the tanks.

A breakdown of airframe weight is shown in Table III.

Propellant Tank Insulation - A satisfactory comparison of insulations for space vehicles can be made on the basis of the conductivity-density parameter, $(k\rho)^{1/2}$ which is the square root of the product of thermal conductivity and density. Both a low density and a low thermal conductivity are desirable, hence, a low $(k\rho)^{1/2}$ value provides the lightest system (assuming no problems of application, etc.). On this basis, multiple foil insulation can be about one-eighth as heavy as evacuated powder insulation and about one-fifteenth the weight of foam insulations for the same protection. The conductivity-density parameter for typical insulations of multiple foil is approximately 0.01.

A typical foil material will have a thermal conductivity of $2 \times 10^{-5} \frac{\text{Btu-ft}}{\text{hr-ft}^2\text{-}^\circ\text{R}}$ between room temperature and liquid hydrogen temperature and a density of 3 to 5 pounds per cubic foot. The primary purpose of the multiple-foil insulation is to reduce the heat transfer due to radiation. The insulation consists of aluminum foil separated by low thermal conductivity spacers (e.g., Linde Superinsulation). The aluminum foils act as radiation shields where the spacers act as separators to prevent adjacent foils from thermally shorting out. In order for the foil pack to be efficient it must operate under a vacuum of 0.1 microns or less to eliminate heat transfer by gaseous conduction.

Another form of multiple-foil insulation is that recently put out by NRC (National Research Corporation). A typical NRC insulation consists of layers of 0.0025 inch mylar sheets. Each sheet is coated on one side with 0.000001 inches of aluminum. It has a k factor roughly equivalent to, or a little less than that of Linde foil. This insulation must also operate under a good vacuum (0.1 microns or less).

The vacuum requirement for efficient operation results in a serious operational problem with foil insulation for cryogenic tankage. Once in space this problem is alleviated. However, while in the atmosphere some means must be provided to maintain the insulation under a vacuum environment. The other alternative is to purge the insulation with a non-condensable gas and then evacuate once in space.

One obvious means of providing a vacuum environment for the insulation to build a double walled tank, but this would incur a large weight penalty.

Another method is to build a light weight flexible vacuum jacket. Such a solution could be provided by a sealed mylar bag placed around the insulated tank and then evacuated. The atmospheric compression load on the insulation would cause higher boil-off while in the atmosphere, however the compressive load would decrease once in the vacuum of space. Linde has suggested the mylar bag vacuum jacket concept and has experimentally investigated the effect of compressing the foil insulation under an atmospheric load. It was found that the heat transfer rate through the compressed insulation under a load of 14.7 psi was roughly 100 times that of the uncompressed insulation at temperatures between 520°R and 37°R. Once the compression load was removed the heat transfer rate approached its original uncompressed value. This simulates a tank being moved from the ground where the evacuated insulation pack would be under an atmospheric pressure load, into space where the compression load no longer exists. The relatively high boil-off rates on the ground could be tolerated if the tanks were continuously topped.

Evacuation of a non-condensable gas from the insulation when in space has been investigated analytically and shows promise. With this method, the foils around cryogenic tankage also would be enclosed in a mylar bag. However, instead of evacuating the foil pack gaseous helium would be introduced at a pressure equal to or slightly higher than atmospheric pressure. Since helium is a gas at liquid hydrogen temperatures, there would be no problem of condensation. During boost, as the surrounding pressure is decreasing, the helium would be allowed to escape through many small holes in the foils. The vacuum of space would then serve as a pump for evacuation of the foils. Since gaseous helium is a relatively good conductor ($k = 0.06 \frac{\text{Btu-ft}}{\text{hr ft}^2\ ^\circ\text{R}}$ between 520 and 37°R) high boil-off losses are incurred whenever helium is present in the foils. Hence, insulation systems which require a long-time period for evacuation would consequently suffer high boil-off losses. With proper design procedures these losses could be minimized.

The means by which foils are applied to flight weight cryogenic tanks could be another problem area. However, because little effort has been devoted to this area more development work is needed to define the severity of the problems. In order for the foils to operate efficiently they must be in a non-compressed state. This means that tightly wound bands used to hold the insulation on the tank would cause undue boil-off. One means of avoiding this might be to use a loose form fitting mylar bag. Additional wire cage support could be used on the bottom of the tank for support during boost.

Another problem area which directly influences the weight penalties associated with propellant storage is that of zero gravity heat transfer. Limited data available from zero-g experiments flown on Aerobee rockets (Reference 5) indicate that propellant temperature stratification will be present. Stratification would cause a non-vented tank to reach some limiting pressure more rapidly (and with less heat input) than a tank with no stratification (i.e., uniform temperature throughout). If no stratification were present, advantage could be taken of the large heat sink available in the propellant and less weight would be required to store the propellants for a given mission.

The insulation concept used in this study was the helium purged foils and the assumptions used in calculating insulation and boil-off weights were:

1. The service module was randomly oriented during the mission;
2. All heat absorbed within the tanks went into vaporizing propellant. No advantage was taken of the heat sink capacity of the propellant. This assumption is necessary since so little is known about zero-g heat transfer;
3. The handbook value of the insulation density was doubled to account for practical application problems;
4. It was assumed that a helium pressure of one micron remained in the foils until the vehicle reached lunar orbit. The foil thermal conductivities used were $7 \times 10^{-4} \frac{\text{Btu-in}}{\text{Hr-Ft}^2\text{-}^\circ\text{R}}$ until arrival at lunar orbit and $2.4 \times 10^{-4} \frac{\text{Btu-in}}{\text{Hr-Ft}^2\text{-}^\circ\text{R}}$ thereafter.

The resulting optimum foil thicknesses were 0.50 inch on the hydrogen tank and 0.65 inch on the LOX tanks. The propellant boil-off throughout the mission is illustrated in Table IV.

<u>Pressurization System</u> - A simple cold helium pressurization system was chosen for this study. The helium was stored in the hydrogen tank at 2000 psia and 38°R. It was assumed that the helium expanded isothermally in the helium bottle and entered the propellant tanks at the propellant temperatures.

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The cold gas pressurization system is heavy and therefore is not necessarily optimum for the cryogenic service module. It was chosen because of its inherent simplicity and should be capable of high reliability.

The cold gas system tends to be heavy because tank venting leads to loss of helium in addition to loss of hydrogen, and helium gas and its associated containment vessel are heavier than hydrogen and its associated tank. Furthermore, it was assumed that during propellant expulsion, no vaporization of propellant into the ullage space took place. This increases the pressurant requirements, but experimental data obtained in Lewis programs substantiate this assumption. After each expulsion period it is assumed that the propellant, pressurant, and propellant vapor come to thermal equilibrium. This necessitates increased venting to prevent overpressure in the tanks. It it were possible to maintain saturation conditions in the ullage space during expulsion, then the pressurant requirements and venting losses could both be reduced substantially.

The utilization of a hot gas pressurization system would reduce the pressurant requirements, but would increase the complexity of the system and result in increased boil-off rates. Similarly, if highly reliable, long life boost pumps were developed, the pressurant requirements could be reduced drastically, but this must be traded off against the weight of the boost pump system and its complexity. If peroxide drive were used, then thermal conditioning of the peroxide would be necessary.

Propellant Flow System - A flow system schematic is shown in Figure 1 along with a definition of schematic notations. All large valves will be operated by intermediate pressure (400 psia) helium. Ullage control will be provided by the attitude control system and it has been assumed that ullage bubble location during zero-g coast can be predicted so that no liquid is vented overboard. In the event that ullage bubble location is not practical, then liquid-vapor separators would be required.

Meteoroid Considerations - In computing the probability of meteoroid penetration, Whipple's 1963A flux distribution and Summers' penetration criteria were used together with a particle density of 0.44 gm/cm^3 and an average velocity of 22 km/sec. These assumptions are based on limited satellite data obtained to date (Reference 6).

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A factor of five was applied to account for the bumper effects resulting from the multilayer airframe configuration. The calculations indicate that the cryogenic service module, as designed, has a 0.999 probability of no meteoroid penetration thru the main propellant tanks for the exposure and conditions of a normal lunar trip. As a result, no additional meteoroid shielding was added.

Propulsion - A service module configuration using a single, gimbaled RL-10 engine was selected. For the LOR Apollo mission, this 15,000 pound thrust engine provides adequate thrust to minimize gravity losses. A specific impulse of 430 seconds at the present RL-10 area ratio of 40 to 1 was assumed.

Use of the present service module attitude control system was assumed for the cryogenic stage. The attitude control propellant requirement and system weight were taken from Reference 4. Additional propellant was added to the existing system to provide for a midcourse correction capability as will be discussed.

To establish pressurant and restart requirements, a schedule of RL-10 propulsion phases was established. Table VI summarizes the number of propulsion phases assumed and the resultant burning times and propellant consumptions for a 90,000 pound lunar transfer weight. Briefly, four midcourse corrections on the outbound trip and four on the return trip were assumed. Three maneuvers were assumed in the vicinity of lunar orbit. One to acquire lunar orbit, one to depart from lunar orbit and one to perform a possible rescue-rendezvous maneuver while in lunar orbit. The number of maneuvers and the ΔV requirements (Table I) leading to the propulsion requirements shown in Table VI were selected somewhat arbitrarily, based on limited Apollo data available at Lewis. For example, a successful rescue-rendezvous maneuver could require more than one service module propulsion phase. However, present Apollo plans for a possible rescue-rendezvous maneuver are not known, and an independent analysis of this maneuver has not been conducted at Lewis. For this maneuver, it was assumed that the major ΔV requirement would be supplied by using the main propulsion system. Additional corrective, trim or closing maneuvers have not been accounted for.

As shown in Table VI, the major midcourse maneuvers were accomplished using the main engine. For the smaller midcourse corrections, when the burning time with main propulsion would be unduly short, four of the 100 pound thrust attitude control engines were used. A comparable thrust level could be achieved by operating the RL-10 engine in the idle mode. Experimental programs at Lewis and at Pratt & Whitney have demonstrated, the feasibility of this mode of operation. However, definitive specific impulse values for the idle mode have not been obtained.

As shown in Table VI, use of the service module attitude control engines for the smaller midcourse corrections can require burning times approaching 100 seconds. During this period attitude control of the service module, is of course, required. In principal, this could be provided by simultaneous use of the service module attitude control engine for midcourse and attitude control, or independently by the command module attitude control system. Since no detailed information is prsently available at Lewis on the capabilities of the present service module and command module attitude control systems, this problem has not been examined in detail.

Total propellant requirements established for the cryogenic service module included provision for chill-down, shut-down, and leakage losses. Chill-down losses were taken as 15.5 pounds per firing, and shut-down losses as eight pounds per firing. Leakage rates were assumed to total 0.0525 pounds per minute for all coast periods after the first firing. These values are consistent with those reported in Reference 7.

Spacecraft Configuration - A layout of the cryogenic service module configuration is shown in Figure 2a. Figures 2b and 2c include some pertinent structural design details. Principal features of the design are a maximum diameter of 192 inches at the base section of the module, a spherical hydrogen tank, a spherical helium tank located inside the hydrogen sphere, and four spherical oxygen tanks. Booster loads are carried through the outer sandwich shell surrounding the stage and the propellant tanks are supported from this shell. The outer shell also serves as a meteoroid bumper. The main propulsion engine thrust load is transmitted through a cross beam and ring arrangement to the outer shell. A combination Fiberglas and foam heat shield and meteoroid bumper is provided across the base of the module. The design is based on a useful propellant loading of 30,000 pounds and a weight breakdown is included in Table VII.

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RESULTS AND DISCUSSION

The improvement in performance occasioned by the introduction of a cryogenic service module into the Apollo mission could be exploited by increasing command module weight, by increasing LEM gross weight, or by reducing required lunar transfer weight. The improvements could also be exploited by providing more ΔV capability in the service module propulsion system in order to increase propellant reserves, or to allow for more energetic rendezvous and rescue operations. Combinations of improvements in all of these areas could also be realized.

In the present study, the advantages of a cryogenic service module have been directed toward increases in LEM gross weight and toward reductions in lunar transfer weight. The results are summarized in Figure 3, where LEM gross weight is plotted as a function of lunar transfer weight for storable and cryogenic service modules. Service module propelland requirements are included in the figure. Performance values indicated for the storable service module are consistent with the ΔV requirements used for the cryogenic service module and the systems weights were taken from Reference 4. The solid curves represent a propellant capacity of 45,000 pounds for the storable service module and 33,000 pounds for the cryogenic service module. Variations in lunar transfer weight are then accommodated by off-loading these designs to obtain a propellant load consistent with the transfer and LEM weight in question. Hardware weights remain fixed along these curves, representing a single design. In the case of the cryogenic service module, increased pressurant requirements were calculated for the off-loaded conditions to account for the greater ullage volumes experienced. In contrast, the dotted curve for the cryogenic stage represents a redesign at each point in question so that hardware weight varies and pressurant is conserved. The weight breakdown of Table VII refers to a design with a 30,000 pound propellant capacity and is consistent with a lunar transfer weight of approximately 90,000 pounds.

Figure 3 demonstrates very little difference in performance between the design-point and fixed cryogenic stages. This is largely the result of the modest ΔV required of the service module in the Apollo mission. Advantages in propellant slosh suppression during boost and in increased volumes available for auxiliary equipment may accompany the smaller designs,

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however. The results indicate that a 90,000 pound transfer weight results in a LEM weight of 30,080 pounds in the case of the storable service module and 40,000 pounds in the case of the cryogenic. At 30,080 pounds LEM weight, the cryogenic stage could accommodate a lunar transfer weight of less than 80,000 pounds. In effect, the cryogenic service module can provide increases in LEM weight of 33 percent or reductions in lunar transfer weight of 14 percent or combinations of improvements in these two areas.

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TABLE I

ASSUMED VELOCITY INCREMENTS

FOR

APOLLO LUNAR ORBIT RENDEZVOUS MISSION

EVENT	Time, hrs.	Velocity Increment, fps	Propulsion
Launch	0		
Insertion	4.5		
Midcourse (1)	13.5	255	RL-10
Midcourse (2)	19.5	60	RL-10
Midcourse (3)	48.5	5	4-100# Att. Cont. Engs.
Midcourse (4)	73.5	10	4-100# Att. Cont. Engs.
Retro into Lunar Orbit	76.5	3553	RL-10
Rendezvous	82.5	500	RL-10
Departure from Orbit	124.5	3971	RL-10
Midcourse (5)	133.5	255	RL-10
Midcourse (6)	139.5	60	4-100# Att. Cont. Engs.
Midcourse (7)	168.5	5	4-100# Att. Cont. Engs.
Midcourse (8)	193.5	10	4-100# Att. Cont. Engs.

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TABLE II

MAJOR DESIGN CONSTRAINTS

Propellant System

LH_2 - Gas Ullage = 5%   Liquid Outage = 1%
LOX - Gas Ullage = 3%   Liquid Outage = 1/2%
LH_2 - Loaded at 17 psia & 38°R   LOX - Loaded at 17 psia & 165°R
Helium pressurant stored at 2000 psia & 38°R
LH_2 Tank max. working pressure - 25 psi at 38°R
LOX Tank max. working pressure - 30 psi at 165°R
Propellant tank insulation - multi-layer foil
  Contained in plastic bag - purged with helium on launch pad

Materials and Allowable Stresses

- Propellant Tanks
  - 2219-T87 Aluminum alloy
  - Allowable stress = 58,000 psi - LH_2 Tank
  - Allowable stress = 54,000 psi - LOX Tank
- Helium Pressurant Tank
  - 5 AL - 2.5 SN Titanium Alloy
  - Working stress = 120,000 psi
- Structure
  - Airframe
    - Aluminum Honeycomb Sandwich
      - 7075-T6 Face Sheets
      - 5052 Honeycomb core
      - Designed on basis of stability
  - Tank Supports
    - 5 AL - 2.5 SN Titanium
    - Designed on basis of stability
  - Thrust Structure, Rings and Longitudinal Stiffeners
    - 7075-T6 Aluminum
    - Working stress = 54,000 psi

Propulsion

Single RL-10 A-3 Engine
P_c = 300 psia
$\mathcal{E}$ = 40:1
I_{sp} (Vacuum) = 430 Seconds
O/F = 5.0
NPSP (Min.)
  LH_2 = 8 psi
  LOX = 13 psi

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TABLE III

AIRFRAME WEIGHT SUMMARY

Outer Shell		1097 lbs.
Face Sheet	348 lbs.	
Honeycomb Core	220	
Longitudinal Stiffeners	120	
Rings	409	
Hydrogen Tank Support		90
Cone	69	
Ring	21	
LOX Tank Support & Thrust Structure		498
Spider Beam	267	
Rings	163	
Cylindrical Shells, Brackets	68	
Helium Tank Support		51
Cone	39	
Rings & Doublers	12	
Aft Heat & Meteoroid Shield		112
TOTAL STRUCTURAL WEIGHT		1848

[redacted]

TABLE IV

PROPELLANT BOILOFF

Propellant Capacity 30,000 lbs.   Tanks Fully Loaded at Launch

EVENT	Time, Hrs.	Ullage Volume, % LOX	Ullage Volume, % LH_2	Boiloff, lbs. LOX	Boiloff, lbs. LH_2
Launch	0	3	5		
Acquire Park Orbit	0.1				
				16	10
Insertion	4.5				
				32	20
Midcourse (1)	13.5				
		9	10	57	56
Midcourse (2)	19.5				
Midcourse (3)	48.5				
		11	13	104	138
Midcourse (4)	73.5				
Retro into Lunar Orbit	76.5				
		72	69	71	64
Recovery	82.5				
		76	74	54	60
Depart Lunar Orbit	124.5				
		97	98	34	32
Midcourse (5)	133.5				
		100	100		
Midcourse (6)	139.5				
Midcourse (7)	168.5				
Midcourse (8)	193.5				

[redacted]

TABLE V

PRESSURANT REQUIREMENTS

Propellant Capacity 30,000 lbs. Tanks fully loaded at Launch

EVENT	Time, Hrs.	Tank Pressure, psia LOX Propulsion	LOX Coast	LH_2 Propulsion	LH_2 Coast	Pressurant Flow, lbs. LOX Tank	Pressurant Flow, lbs. LH_2 Tank
Launch	0	17	17	17	17		
Acquire Park Orbit	0.1	17		17			
Insertion	4.5	17		17			
Midcourse (1)	13.5	30		25		2	17
Midcourse (2)	19.5	30		25		1	8
Midcourse (3)	48.5	17		17			
Midcourse (4)	73.5	17		17			
Retro into Lunar Orbit	76.5	30		25		16	189
Recovery	82.5	30		25		7	44
Depart Lunar Orbit	129.5	30		25		11	119
Midcourse (5)	133.5	30		25		8	50
Midcourse (6)	139.5	17		17			
Midcourse (7)	163.5	17		17			
Midcourse (8)	193.5	17		17			

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TABLE VI

PROPULSION REQUIREMENTS

EVENT	Time,Hrs.	Propulsion	Propellant Required,lbs.	Burning Time,Sec.
Launch	0			
Insertion	4.5			
Midcourse (1)	13.5	RL-10	1641	47.1
Midcourse (2)	19.5	RL-10	381	10.9
Midcourse (3)	48.5	4-100# Att.Cont.Engs.	43	34.0
Midcourse (4)	73.5	4-100# Att.Cont.Engs.	87	67.7
Retro into Lunar Orbit	76.5	RL-10	19705	564.9
Rendezvous	82.5	RL-10	935	26.8
Depart Lunar Orbit	124.5	RL-10	6232	178.7
Midcourse (5)	133.5	RL-10	337	9.7
Midcourse (6)	139.5	4-100# Att.Cont.Engs.	107	83.8
Midcourse (7)	168.5	4-100# Att.Cont.Engs.	9	6.9
Midcourse (8)	193.5	4-100# Att.Cont.Engs.	18	13.7

[redacted]

[redacted]

TABLE VII

CRYOGENIC SERVICE MODULE

WEIGHT BREAKDOWN

Helium	566 Lbs.
Helium Tank Shape-Sphere Diameter - 57.4" Thickness - 0.239 " Material-TI-5AL-2.5SN	398
Fuel Tank Shape - Sphere Diameter - 167.0" Thickness-0.020" (Minimum Gage) Material-2219-T87AL	187
Oxidant Tanks Shape - Spheres (4) Diameter - 67.8" Thickness - 0.015" (Minimum Gage) Material - 2219-T87AL	89
Main Engine	290
Propellant System	200
Structure	1848
Attitude Control System	1680
Insulation	413
Impulse Propellant	30000
Boil-off, Residuals, Leakage & Chilldown Propellant	1595
Contingency	1068
STAGE TOTAL	38334

[redacted]

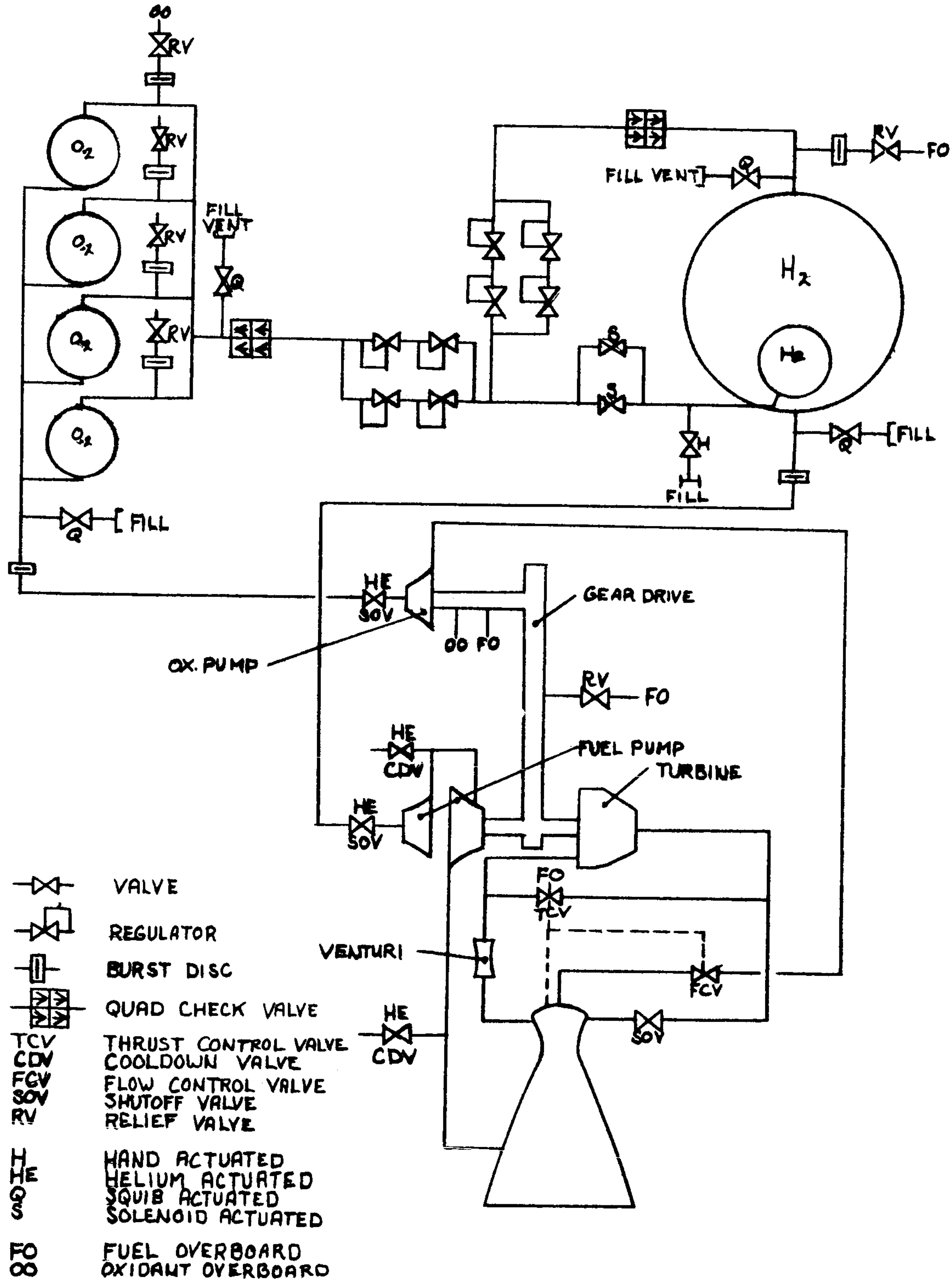


FIGURE 1 FLOW SYSTEM SCHEMATIC FOR CRYOGENIC SERVICE MODULE

[redacted]

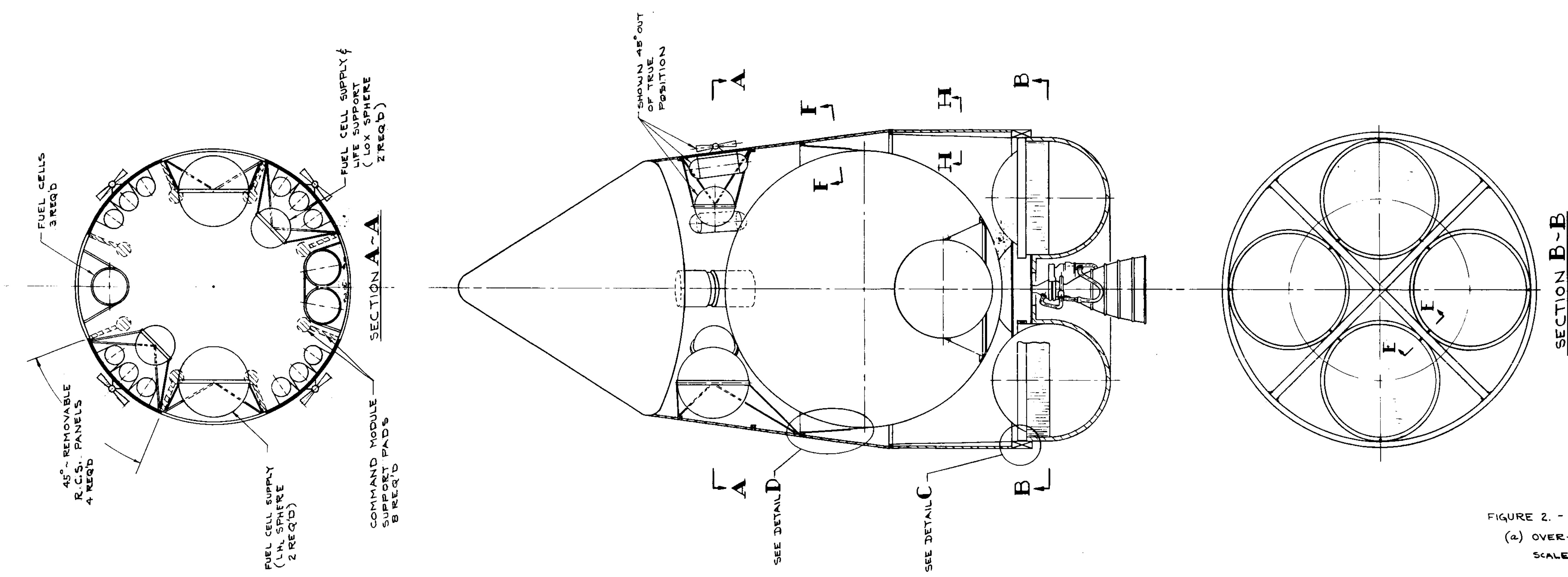


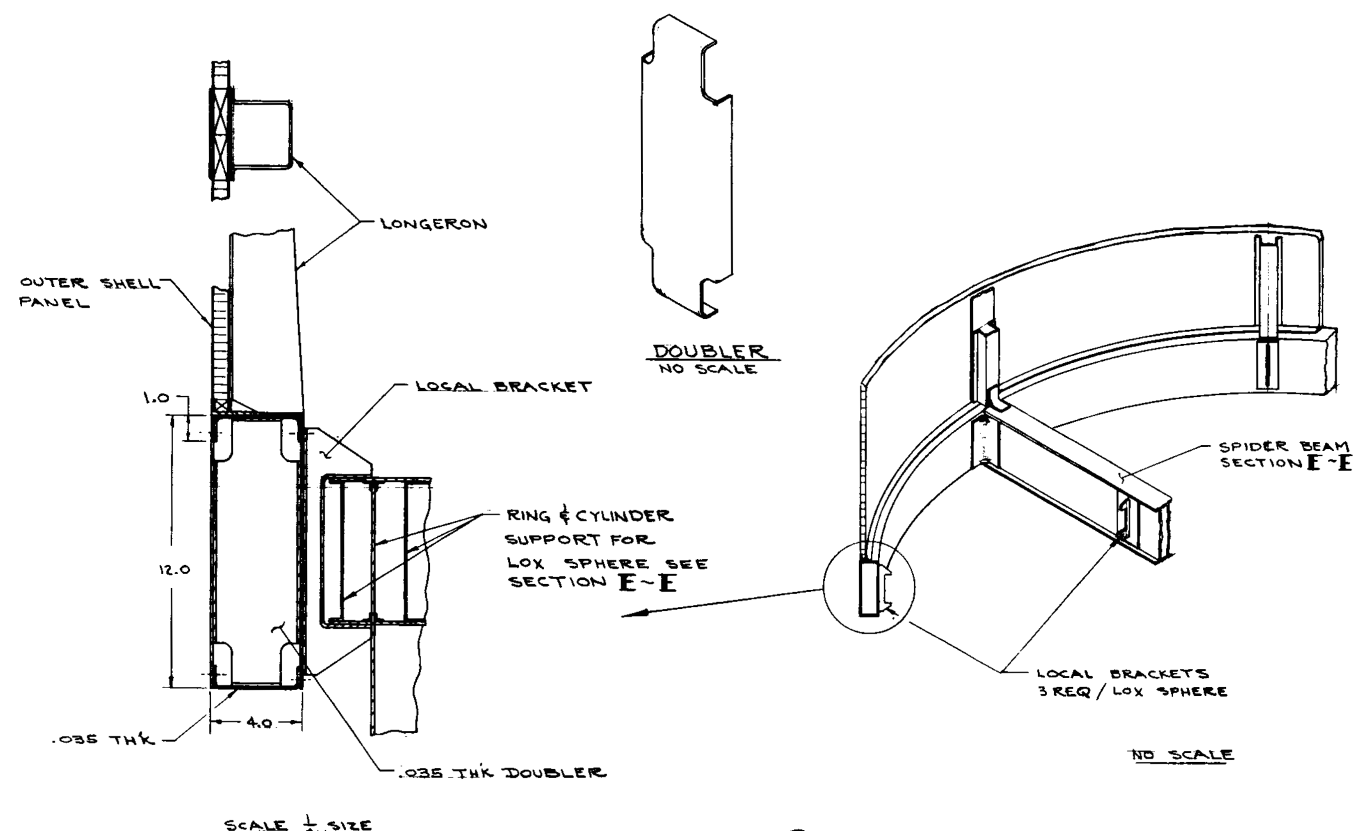
FIGURE 2. - CRYOGENIC SERVICE MODULE

(a) OVER-ALL CONFIGURATION

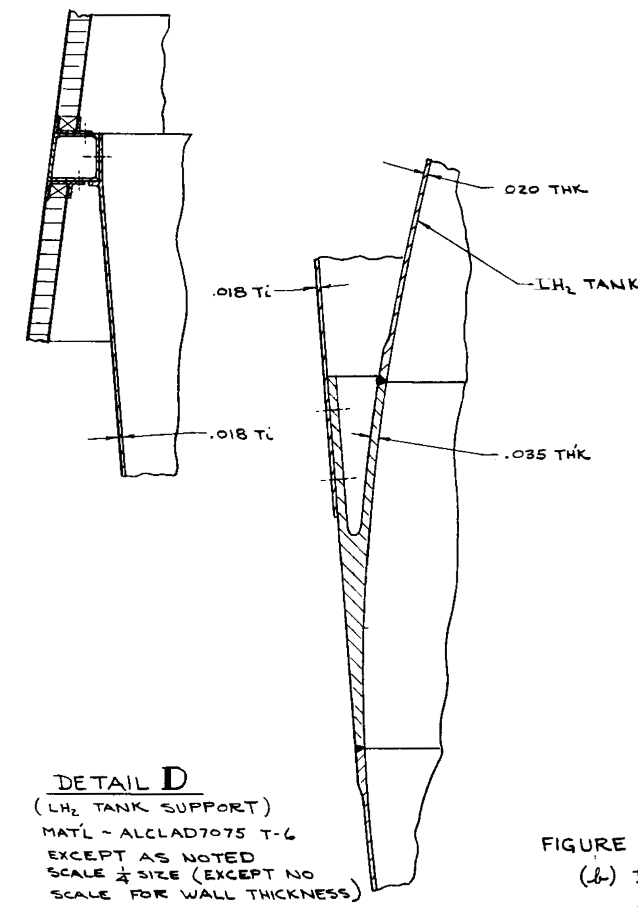
SCALE 3/8" = 1'-0

[redacted]

[redacted]



DETAIL C
MAT'L ~ ALCLAD 7075 T-6
EXCEPT AS NOTED

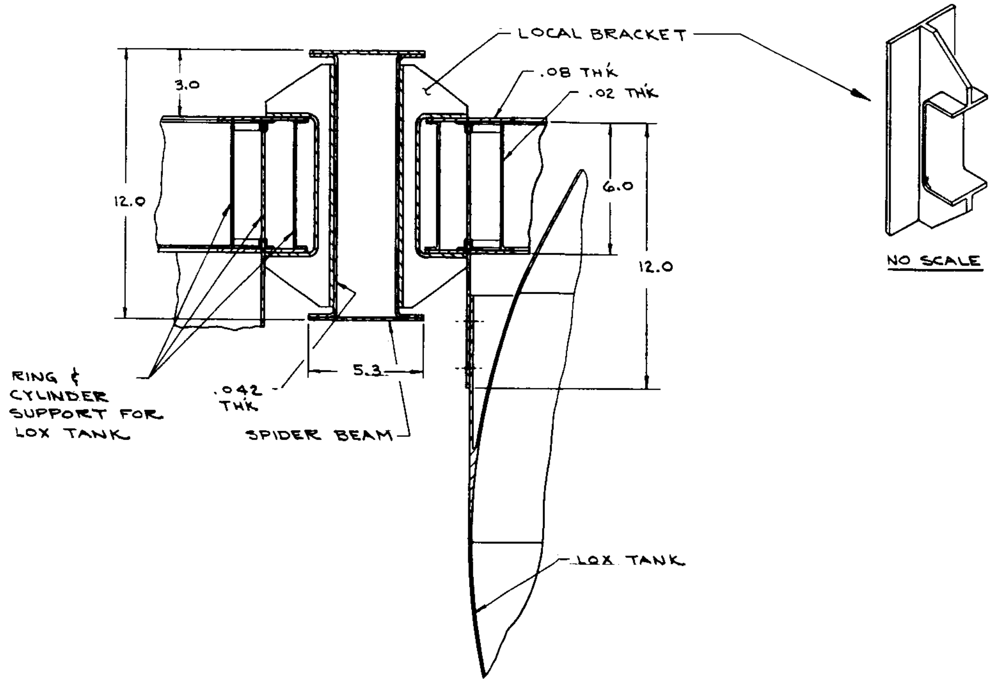


DETAIL D
(LH_2 TANK SUPPORT)
MAT'L ~ ALCLAD7075 T-6
EXCEPT AS NOTED
SCALE 1/4 SIZE (EXCEPT NO SCALE FOR WALL THICKNESS)

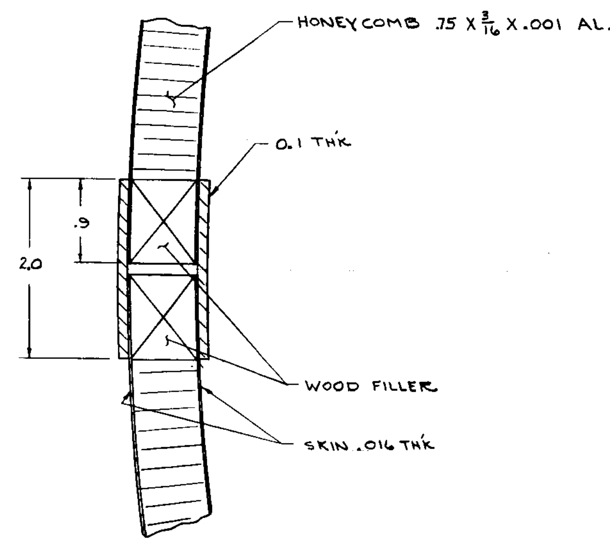
FIGURE 2 - CRYOGENIC SERVICE MODULE (CONT.)
(b) DESIGN DETAILS
SCALE AS NOTED

[redacted]

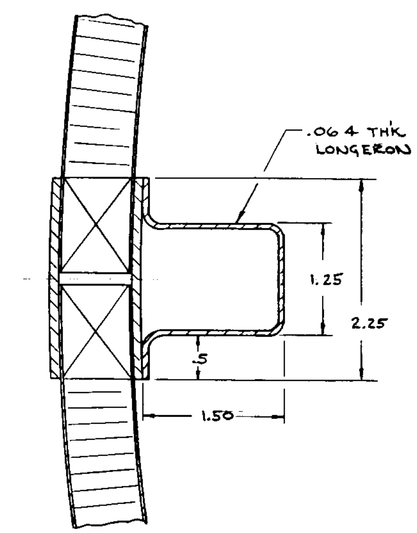
[redacted]



SECTION E~E
(SPIDER BEAM & LOX TANK SUPPORT)
MAT'L~ALCLAD 7075 T-6
SCALE $\frac{1}{4}$ SIZE



SECTION F~F
TYPICAL SKIN PANEL
SPLICE & SHELL CONSTRUCTION
MAT'L ~ ALCLAD 7075 T-6 EXCEPT AS NOTED
SCALE FULL SIZE



SECTION H~H
TYPICAL LONGERON & SPLICE CONSTRUCTION
MAT'L ALCLAD 7075 T-6
SCALE FULL SIZE

FIGURE 2 - CRYOGENIC SERVICE MODULE (CONT.)
(c) DESIGN DETAILS
SCALE AS NOTED

[redacted]

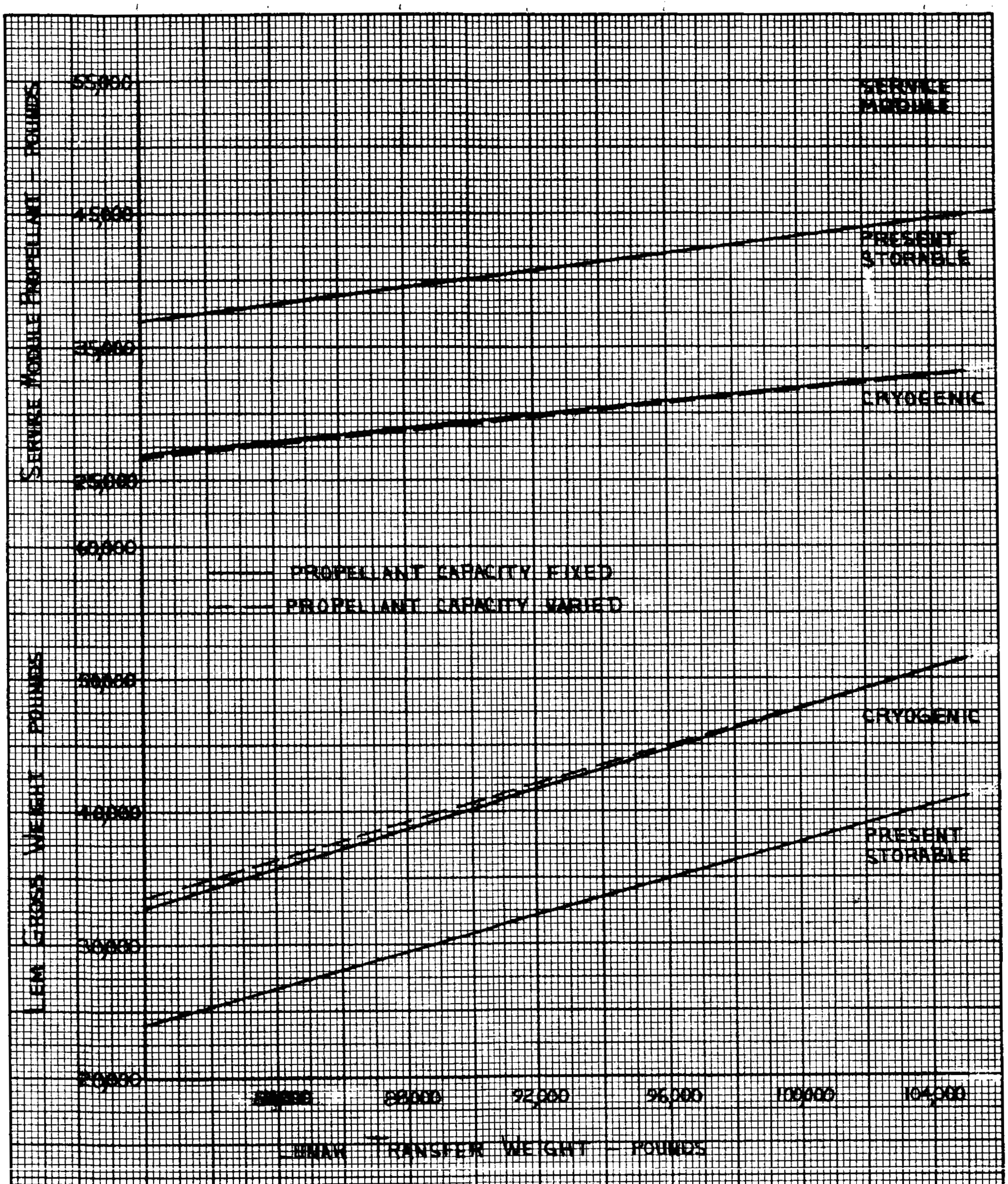


FIGURE 3 Capability of Present Service Module Compared With Cryogenic Service Modules